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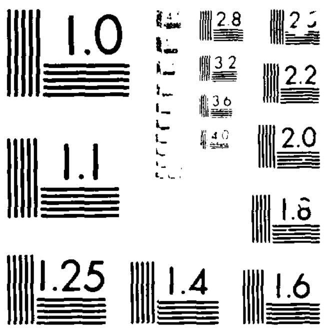
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State-of-the-Art Review on Composite Material Fatigue/ Damage Tolerance

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December 1985

Final Report

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16. Abstract

A state-of-the-art review on composite material fatigue/damage tolerance was conducted to investigate the literature for fatigue life prediction methodologies including stress-based methodologies, strength degradation models, and damage growth models. A critical review was made of each methodology and its commensurate basic equations of importance. Experimental data were reviewed and the behavior of specimens was correlated with that of civil aircraft components. The report also examined the six recognized methods for the non-destructive testing of fibrous composite materials and identified the most effective methods.

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PREFACE

This Final Report was prepared by B & M Technological Services, Inc. (B & M), Cambridge, Massachusetts, and its subcontractor, Atlantic Science and Technology Corporation, Cherry Hill, New Jersey, for the Federal Aviation Administration Technical Center, Atlantic City, New Jersey. B & M and AS&T wish to thank the Technical Officer, Mr. Lawrence M. Neri, for his suggestions and guidance throughout the course of this work. Particular thanks are also extended to Mr. Caesar Caiafa, Branch Manager of the Crashworthiness/Structural Airworthiness Branch, for his experience and professional insight in the broad range of composite structures.

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EXECUTIVE SUMMARY

The state-of-the-art review on Composite Material Fatigue/Damage Tolerance has served to direct more attention toward the complexities involved in identifying accurate theories and methods needed to characterize fatigue failure criteria and, in particular, to identify fatigue failure mechanisms.

An extensive literary survey was conducted to compile references related to the general area of fatigue of composite materials and, in particular, to identify those references which are directly related to fatigue/damage tolerance. While the mechanical and physical properties of laminae are well known, the precise determination of fatigue failure characteristics of a laminate is quite complex. Presently, there is no precise definition regarding what constitutes fatigue damage of the overall laminate. However, accumulated fatigue damage of the various laminae taken on a collective basis forms the definition of laminate fatigue damage. More specifically, while the fatigue failure of the laminate is identified by being able to predict, analytically, the fatigue failure of a uniaxial fiber composite, two problems remain regarding the analysis of fatigue failure of unidirectional fiber composites. Namely, there is no established fatigue failure criteria for combined cyclic stresses and the inherent difficulty in predicting lifetime under variable amplitude cycling - which is known as the cumulative damage problem. Consequently, the complexities associated with fatigue/damage tolerance are augmented by the fact the fatigue failure process encompasses initiation, development, and termination aspects of failure. The failure life prediction methodologies constitute fundamental ways to measure fatigue in fiber composite structures. Presently, these methodologies include stress-based methodologies, strength degradation models, and damage growth models.

Stress-based methodologies are characterized by the fact that there are certain fundamental laminae stresses, found through measurement or calculation, which can be linked to the damage states of the laminae and subsequently used to predict the fatigue life of the laminate. The study provided a critical review of stress-based methodology literature and found a link between static strength and a basic cumulative damage model. The form of the failure function is found through S-N curves, found from testing off-axis unidimensional specimens. The failure criterion, in turn, is based on information manifested through the material fatigue functions and the stress ratio.

Strength degradation models are concerned with concepts manifested in the prediction of lifetime under cyclic loading conditions. Residual strength degradation is used as a measure of fatigue failure of fiber composites. When static strength, defined as residual strength after N cycles, equals the maximum stress amplitude, fatigue failure occurs. While the concept of residual strength remains an important parameter in the identification of lifetime for a unidimensional fiber composite, strength degradation of laminates is a complex problem and presently cannot be defined or evaluated strictly in terms of macromechanics. Thus, stiffness reduction is used to measure laminae damage accumulation.

Regarding damage growth modeling, this study indicates that a number of researchers emphasize that delamination growth is the fundamental issue in the evaluation of laminated composite structures for durability (fatigue) and damage tolerance (safety). However, there are pitfalls associated with fatigue damage modeling, and more attention must be paid to the number of cycles to failure, particularly regarding the scale of the body being modeled.

Regarding non-destructive testing methods of fibrous composite materials, there are presently six recognized approaches, however, no particular non-destructive technique can be used with certainty for all configurations. Test methods must be selected and tailored to each item, and the geometry of the part must also be taken into account.

For inspection of damage after dynamic fatigue loading, ultrasonic techniques should be included among the most useful methods. On the other hand, holography is effective as far as detecting delaminations and cracks in the surface piles but does not detect subsurface matrix cracking. Likewise, since video thermography relates the thermal patterns more directly to the stress field in the material, it is a more appropriate model for studying the mechanical behavior of composite materials.

INTRODUCTION

BACKGROUND AND OBJECTIVES.

The investigation of fatigue/damage tolerance problems during the past five years has generated a significant amount of information, both analytical and experimental, pertaining to composite materials. Many new composite materials have been developed during this period -- so many, in fact, that the government and the industrial world are looking for ways to qualify and quantify the mechanical behavior exhibited by combinations of materials heretofore not used.

At the present time, numerous procedures, some analytical - others empirical, are used to substantiate the fatigue/damage tolerance aspects of civil and military composite aircraft structures. However, no single or series of widely-recognized procedures exist for verifying the basic fatigue mechanisms associated with fatigue/damage of these structures.

Fiber reinforced materials such as carbon, graphite, boron, and glass reinforced plastics are currently certified for usage in composite aircraft structures. On the other hand, new epoxy resins are currently being studied for usage in composite aircraft structures.

The objective of this study is to perform a comprehensive quantitative analysis of fatigue/damage tolerance methodologies of composite materials and correlate these methodologies with empirical data in order to establish a procedure for evaluating composite materials used in civil aircraft structures.

LITERARY SURVEY

Although the mechanical and physical properties of laminae are now well known, the precise determination of fatigue failure characteristics of a laminate is quite complex. More analytical work remains to be done before this particularly thorny problem can be considered solved (reference 1).

In essence, a characterization (or characterizing) of a composite material, in regard to fatigue, is simply a description of characteristics or peculiar qualities. Thus, one can focus attention on identifying fatigue failure processes. Hashin (reference 1) indicates there are two major failure processes: intralaminar and interlaminar. In the intralaminar fatigue failure process, the intralaminar cracks which have accumulated in the fiber or matrix modes run parallel to the fibers. The interlaminar fatigue failure process involves the opening up of an interlaminar edge crack which splits the laminate, but continues to grow as the cycling process continues.

The question of fatigue damage for laminates is even more complex, and it is necessary to define fatigue damage, at least in some qualitative form, before the notion of fatigue life prediction can be addressed. Fong (reference 2) in his definitive paper uses four typical damage parameters to identify damage. They include normalized residual tensile strength, maximum damage length, number of debonded fibers, and total resin crack length. He concludes that the last damage parameter shows the greatest promise regarding the identification of fatigue damage on an analytical, rational basis. While the precise question regarding what constitutes fatigue damage of the overall laminate has not been answered, it is safe to say the accumulated fatigue damage of the various laminae taken on a collective basis forms the definition of laminate fatigue damage. Thus, presently, the question of fatigue failure of the laminate is answered by being able to predict, analytically, the fatigue failure of a uniaxial fiber composite. Hashin (reference 1) indicates the two major problems in analysis of fatigue failure of undirectional fiber composites are: (1) establishment of fatigue failure criteria for combined cyclic stress; and (2) prediction of lifetime under variable amplitude cycling - which is known as the cumulative damage problem. Hence, the fatigue failure process encompasses initiation, development, and termination aspects of failure. The fatigue life prediction methodologies constitute reasonably fundamental ways to measure fatigue in fiber composite structures. Presently, these methodologies encompass three fundamental types, namely: stress-based methodologies, strength degradation models, and damage growth models.

STRESS-BASED METHODOLOGIES.

These methodologies are characterized by the fact that certain fundamental stresses are known (measured or calculated) in the laminae which can be linked to the damage states of the laminae and subsequently used to predict the fatigue life of the laminate.

While experiments alone have not been sufficient to describe the failure behavior and thus provide the foundation of a failure criteria representative of all fiber composites, they did, during the early 1970's, provide valuable information regarding the relationship between failure and fiber-matrix strength and properties. Rosen and Dow (reference 3) conducted experiments which showed a link between static strength and a basic cumulative damage model.

Hashin and Rotem (reference 4) recognized that fatigue failure, due to the extreme complexities involved in attempting to characterize fibrous composite behavior, should be based on macromechanics and macroscopic-oriented criteria, wherein such failure criteria can be identified based on the average stresses to which the composite is subjected. The form of the failure criterion is dictated by two distinct experimentally observed failure modes. Three S-N curves, found from testing of off-axis unidirectional specimens undergoing uniaxial load, are used to express the failure criterion. The failure criterion, in turn, is based on information manifested through the material fatigue functions and the stress ratio.

Mandell and Meier (reference 5) described crack growth in a stepwise fashion with the crack remaining stationary for many cycles before each step of growth. They use the S-N curve of the unnotched material to describe how the ligament at the crack tip is fatigued. Using an assumed stress field and cumulative damage law, the number of cycles for initial growth from a notch and the rate of crack growth are predicted. The experimental results agree well with this simple theory.

Rotem and Hashin (reference 6) used failure criterion (reference 4) to determine if the subsequent fatigue of laminates can be predicted based on the presence of failed or degraded laminae. Using the results of a recent analytical and experimental investigation (reference 7), it was concluded for angle plies greater than 45°, the failure criterion is substantiated by good agreement between theory and experiment, whereas for angle plies less than 45°, the failure criterion underestimated the fatigue failure load.

The effects of compression load on the failure response of fibrous laminates were investigated by Ryder and Walker (reference 8). Extensive testing, under constant amplitude loading, was conducted at three different stress levels. It was found that compressive load greatly reduced fatigue life at lower stress levels.

Angle-ply notched and unnotched fibrous composites were studied experimentally by Ramani and Williams (reference 9). Using unnotched specimens, S-N curves were drawn using various stress ratios R. In particular, tension-tension (T-T), tension-compression (T-C) and compression-compression (C-C) cycling studies were conducted in order to determine fatigue damage. Overall results were expressed in the form of a constant life diagram (Goodman diagram) showing the relationship between mean stress and stress amplitude. Experimental results indicate it is possible to relate notched fatigue behavior to unnotched fatigue behavior for various laminates under T-T cycling. Resistance to damage accumulation under T-C cycling can be effectively compared for various laminates by measuring changes in crack-opening displacement (COD) during cycling.

Sims and Brogdon (reference 10) recognized that when the matrix contributions to load carrying capability are significant, the fatigue characteristics of these composites can be quite different from those of the fiber-dominated matrix. They performed experiments to gain a better understanding of matrix-dominated fatigue behavior of fibrous composites. Existing static strength failure theories were used to predict fatigue strength. These theories require a knowledge of fatigue functions in the principal material directions of a laminae to predict the first-ply failure of a laminate. Using general regression analysis on the fatigue test data at various stress ratios, much of the testing required to develop the S-N curve at specific steady-stress levels was eliminated, and the use of the static based theories reduced the required amount of fatigue testing of laminates composed of different fiber orientations.

Wang, Chou, and Alper (reference 11) investigated experimentally the effects of static proof testing on the statistical distribution of the static strength and fatigue life of a unidirectional laminate. Using this proof-test procedure and unidirectional test data, they verify that the equal-rank assumption appears to be both reasonable and practical. However, additional study is needed to determine the practicality of using this concept for laminates of different fiber orientations and stacking sequences.

Hashin (reference 12) provides conceptual insight into the rationale underlying the establishment of three-dimensional macromechanical static and fatigue criteria for unidirectional fiber composites. Based on significant laboratory data for the static case, it was concluded that such composites exhibit four distinct failure modes. For fatigue failure, there is a family of fatigue criteria, each associated with a different lifetime. Further, it was recognized that transverse isotropy exists in the composite. Analytically, quadratic stress polynomials are used to model fatigue behavior.

STRENGTH DEGRADATION MODELS.

Strength degradation models are concerned with concepts manifested in the prediction of lifetime under cyclic loading conditions. Presently, the most fundamental work has been done on unidirectional fiber composites and, therefore, fatigue lifetime is defined in terms of failure (or cumulative damage) after the composite has undergone N cycles.

Residual strength degradation is used as a measure of fatigue failure of fiber composites. When the static strength, defined as residual strength after N cycles, equals the maximum stress amplitude, fatigue failure occurs. Thus, the concept of residual strength remains an important parameter in the identification of lifetime for a unidimensional fiber composite.

Strength degradation of laminates is a complex problem and presently cannot be defined or evaluated strictly in terms of macromechanics. Hence, laminae damage accumulation is measured in terms of stiffness reduction.

Yang (reference 13) derived a new fatigue residual strength degradation model based on the assumption that the residual strength decreases monotonically. He used the theory of periodic proof tests and the reliability prediction for composites, which assumes a particular residual strength degradation model (references 14-17) for unnotched composite laminates that indicates that the residual strength R(N) after n fatigue cycles is a monotonically decreasing function of N. The resulting fatigue life distribution follows a three-parameter Weibull statistical distribution. Exceptionally good correlation between experimental results and theory was found.

Yang (reference 18) generalized the residual strength degradation model to account for the effect of tension-compression fatigue loading. Again, good correlation was found between theory and statistical distributions of residual strength and fatigue life. Further work by Yang (reference 19) resulted in a three-parameter fatigue and residual strength degradation model to predict statistically the fatigue behavior of composite laminae under axial shear loadings.

Based on a review of extensive fatigue failure information, Hashin and Rotem (reference 20) developed a rational phenomenological theory of fatigue life prediction under arbitrary variation of cycle amplitude. While not specifically oriented toward fiber composites, the damage curves developed helped to establish a cumulative damage theory which could be used to describe the uniqueness of the damage curve.

Chou and Croman (reference 21) developed equations for the distribution of residual strength. Using the strength-life equal rank assumption of Hahn and Kim (reference 22), it was shown that their equations compared well with existing experimental results. The change of residual strength can be of weak degradation, strong degradation, or increase in strength.

Kim and Park (reference 23) investigated the probability of a relationship between static strength and fatigue life. Using two-parameter Weibull distributions, proof testing was conducted at various levels of proof stress to study the effect of proof loading on fatigue life. The experimental results showed, for tension-tension fatigue loading, an excessive proof loading results in premature failure in fatigue.

Whitney (reference 24) developed a procedure that allows the generation of an S-N curve with some statistical value without resorting to an extremely large database. This approach is compatible with wearout or strength degradation. It is recommended that a maximum likelihood estimator (MLE) be used to determine Weibull parameters.

Matrix cracking was the focus of research conducted on composite specimens by Highsmith and Reifsnider (reference 25), since it is recognized that matrix cracking is the source of stiffness change which occurs early in the life of a specimen or component. Building on earlier qualitative studies (reference 26), experiments were conducted to isolate stiffness changes, due to matrix cracking, and create models wherein these changes can be studied analytically. It was found that tensor stiffness changes due to matrix cracking can be predicted using simple lamina stiffness reduction principles and standard laminate analysis.

Quantitative studies on delamination growth and stiffness loss were conducted by O'Brien (reference 27) using information found from research done by Rybicki et al. (reference 28) which characterizes delamination growth based on the rate of strain energy released, G. A simple technique was developed to measure the onset and growth of delaminations in unnotched graphite/epoxy laminates. Using a critical value for shear modulus, G_c, it

... found this particular value may be independent of the ply orientations that make up the delaminating interface. Thus, the delamination resistance curve (R-curve) and power law developed on [$\pm$ 30/$\pm$ 30/90/90]s laminates can be used to predict delamination growth in other laminates.

The two-parameter and three-parameter Weibull distributions are used to form residual strength models for fatigue analysis. However, Whitney (reference 29) was able to overcome many of the disadvantages of Weibull distributions by considering the lognormal distribution for analyzing composite material data. Further, the lognormal distribution also can be used with the wearout model, however, the probability density function should be used instead of the cumulative probability function.

Ratwani and Kan (reference 30) emphasize that for composites, compression-fatigue is more degrading, in terms of life, than tension-fatigue. Using this assumption, a model for predicting compression residual strength of composites subjected to compression-fatigue is developed and verified by test data. The residual strength function is expressed in terms of the static strength and an arbitrary function related to the size of the delaminations produced during N number of fatigue cycles.

Talreja (reference 31) developed a stiffness-based fatigue damage characterization wherein changes in all four independent stiffness constants of an orthotropic elastic lamina are considered. It was found that shear modulus and Poisson's ratio changed significantly.

Variational techniques were used by Gottesman, Hashin, and Brull (reference 32) to study the reduction of elastic moduli of unidirectional fiber composites due to parallel cracks. Equations for upper and lower bounds of effective elastic moduli were developed.

Using acoustic techniques, Holt and Worthington (reference 33) tested CFRP and GFRP specimens during tension-tension cycling. For CFRP specimens, continuous monitoring failed to provide warning of impending fatigue failure. For GFRP specimens, a different damage process occurs for failure. This process can be related to fatigue life.

O'Brien and Reifsnider (reference 34) measured stiffness reductions of unnotched boron/epoxy laminates. Fatigue damage was observed under cyclic tension loading in order to assess: (1) the extent of fatigue damage from measured dynamic stiffness loss; (2) the anisotropy of fatigue damage from changes in the longitudinal stiffness, Eyy, shear stiffness, Gxy, and transverse stiffness, Eyy, using a combination of unaxial tension, rail shear and flexure tests; and (3) the validity of the secant modulus criterion for predicting stiffness loss at failure from static longitudinal stiffness changes measured during fatigue. These results showed fatigue damage consisting of matrix crazing was fairly uniform throughout coupons of [$\pm$45]s laminates. Fatigue damage in [0/90]s laminates was localized, consisting of transverse cracks spaced along the specimen length, but for [0/90/$\pm$45]s laminates, fatigue damage consisted of both localized ply crazing and uniform matrix cracking. After applying tension-tension cyclic loading in the X direction, for [0/90]s laminates, the relative order to stiffness changes was ΔEyy, ΔGxy, ΔEyy. Since damage growth and stiffness loss are load-history dependent, the secant modulus criterion is not valid for general application.

Fundamental fracture considerations were studied by Reifsnider and Jamison (reference 35) in order to assess the manner in which prefracture fatigue damage affects residual strength and the fracture process. It was found that, while distinctive mechansims of damage have been identified and associated with fatigue loading, no mechanistic scheme for associating the rate of damage development with pertinent details of mechanical and material circumstances has been found. Of course, rate equations can be empirically associated with the damage development (as is done with schemes such as the wearout model), but a single characterization of the rate of development of fatigue damage (as defined by Reifsnider and Jamison) in general, based on observed microdamage details and the principles of mechanics, has not been found. Microstrains, due to internal stress redistribution, have verified that the internal stress redistribution due to the types of fatigue damage observed is of the type and magnitude that can explain the observed changes in residual strength of composite laminates.

DAMAGE GROWTH MODELS.

Damage growth modeling essentially identifies a way fatigue damage can be modeled. Wilkens, Eisenmann, Camin, Margolis, and Benson (reference 36) emphasize that delamination growth is the fundamental issue in the evaluation of laminated composite structures for durability (fatigue) and damage tolerance (safety). They cite the work of a number of researchers (references 37-43) who indicate that when test conditions are extended to explore failure mechanisms, delamination is observed to be the most prevalent life-limiting growth mode. Characterization of the behavior of delamination has been approached by adopting and developing techniques for coupon design, static and fatigue testing, data analysis, fracture analysis for separation of modes, spectrum life prediction, and spectrum truncation. Critical strain-energy release-rate values have been obtained for Mode I (tensile opening mode) and Mode II (forward shear mode). The applied cyclic load must be nearly equal to the critical static load to obtain observable growth in the tensile opening mode. But for the graphite/epoxy delamination in the forward shear mode, it is suggested that shear is the chief subcritical growth mode for graphite/epoxy.

Fong (reference 2) has added a note of caution on the pitfalls of fatigue damage modeling. While the goal is to predict the number of cycles to failure (N_f), more attention should be paid to this critical value at the local, specimen, and structure levels. Hence, confusion over scale is a pitfall to which many authors succumb. Other pitfalls involve oversimplification regarding the substitution of linear models for nonlinear models, lack of delamination between two regimes of fatigue cycling, and the lack of proper data acquisition and data analysis.

Ramkumar (reference 44) investigated the effect of imbedded (idealized) delamination on the compression fatigue behavior of quasi-isotropic graphite/epoxy laminates. It was found that the predominant failure mode in the test specimen was the propagation of imbedded delaminations in the tab region.

Experiments by Ratwani and Kan (reference 45) found that stacking sequence had a significant effect on damage growth and failure modes of graphite/epoxy coupons.

A fatigue/damage mechanism was observed by Badaliance and Dill (reference 46) through formulation of a damage-indicating parameter based on the intralaminar microcracking of the resin and its application in conjunction with a linear fatigue/damage model to predict spectrum life of graphite/epoxy laminates. The damage correlation parameter is based on a strain energy density factor. A fatigue damage model by Broutman and Sahu (reference 47) was used to predict spectrum fatigue life.

Sandhu, Gallo, and Sendeckyj (reference 48) employ a progressive-ply-failure finite element program for predicting damage initiation and progression. While this particular finite element method program serves as a viable procedure for predicting the damage progression, attention must be directed toward conducting additional experiments in order to verify the mode. Further, the program should be extended to account for delaminations.

Crossman and Wang (reference 49) conducted tension experiments on graphite/epoxy laminates, recognizing that the process of composite laminate fracture under static or fatigue loading is known to involve a sequential accumulation of damage, in the form of matrix-dominated cracking, prior to final fracture by fiber breakage in the primary load-carrying plies. Using information from studies made by a number of researchers (references 50-53) who found that ply thickness has an effect on damage mode and delamination, studies were made to delineate the degree of structural modeling necessary to predict fracture successfully in composite laminates. It was found that while stress and energy methods prove useful in predicting the onset of transverse cracking, the density of transverse cracking, and the onset of delamination at the laminate free edges, more detailed analysis is necessary for prediction of the saturation density of transverse cracks, delamination growth under fatigue loading, and the ultimate strength of the primary load-carrying plies.

Reddy (reference 54) conducted extensive fatigue testing of coupons, structural elements, and full-scale helicopter blades. S-N curves were developed using data from coupon and element tests. Using statistical analysis to adjust these curves, a revised Miner's cumulative damage method was used to calculate fatigue life. A damage growth test was made of a partially failed blade. These test results substantiated the excellent fatigue and damage growth characteristics of the composite blade.

Oldyrev (reference 55) developed a new method of fatigue testing which permits the fast tests of one specimen to be used to determine the fatigue life of the material for three to six load levels. The proposed method is based on the laws of fatigue/damage accumulation.

Poursartip, Ashby, and Beaumont (reference 56) developed a damage function which can be determined by measuring the changes of modulus with cycling.

Structural models based on continuum-fracture mechanics principles were developed by Bolotin (reference 57). In particular, micromechanics concepts related to crystalline-fiber structure were used to establish equations for the growth of fatigue cracks.

The use of laminate stiffness reduction as a means of interpreting damage was developed by Jamison and Reifsnider (reference 58). Various matrix damage modes were related to the corresponding matrix cracks which formed this damage.

BASIC EQUATIONS OF IMPORTANCE.

During the past ten years, significant theoretical and experimental studies have been conducted on fibrous composite materials. While the basic equations associated with unidirectional fiber composites are important, the problems related to the fatigue/failure mechanisms and the prediction of fatigue/damage and lifetime of a particular laminate are much more complex. In particular, the somewhat random nature of mechanisms forces one to employ statistical theories and means, in addition to macromechanics theories, to generate equations of importance and empirical expressions to explain these important factors.

Hashin (reference 4) indicates there is a family of failure criteria, each associated with a different lifetime, wherein fiber rupture, matrix cracking, and fiber/matrix interface bonding are directly related to the failure process. These manifestations can be defined under the broad category of damage. In turn, damage results in a loss of stiffness and the decrease of residual strength and lifetime during fatigue cycling. The mechanical study of internal flaws is often called damage mechanics. He cites important fundamental qualitatively-oriented work by a number of researchers in the development of a damage accumulation model for tensile failure in the fiber directions when fiber strengths are statistically scattered, and the investigation of compression failure in fiber direction in terms of fiber buckling. He assumes the average stress state

$$F(\sigma_{ij}) = 1 \qquad (1)$$

Using the general quadratic failure criterion proposed by Tsai and Wu

$$F_{ijkl}\sigma_{ij}\sigma kl + F_{ij}\sigma_{ij} = 1 \qquad (2)$$

(reference 59) leads to the description of the failure surface by a single polynomial in the stresses. Using the previous equation for plane stress results in

$$F_{1111} = \frac{1}{\sigma_A^+ \sigma_A^-} , \qquad F_{2222} = \frac{1}{\sigma_T^+ \sigma_T^-}$$

$$F_{11} = \frac{1}{\sigma_A^+} - \frac{1}{\sigma_A^+} , \qquad F_{22} = \frac{1}{\sigma_T^+} - \frac{1}{\sigma_T^-} \tag{3}$$

where $\sigma_A^+, \sigma_A^-, \sigma_T^+, \sigma_T^-$ represent the ultimate stresses in the fiber in the transverse directions.

Failure modes are classified as tensile/fiber/mode, compressive failure mode, and the matrix mode. See figures 1 and 2. Thus, failure criteria for the tensile/fiber and tensile/matrix modes are

$$\left(\frac{\sigma_{11}}{\sigma_A^+}\right)^2 + \left(\frac{\sigma_{12}}{\tau_A}\right)^2 = 1$$

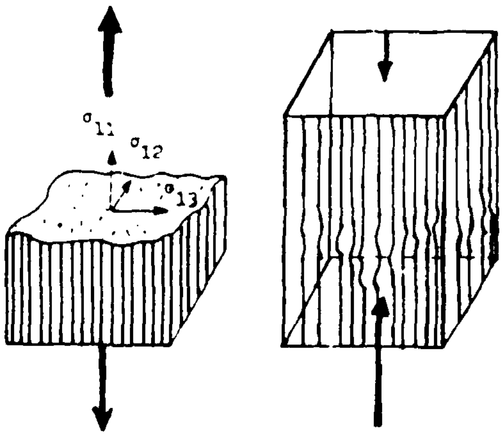
$$\left(\frac{\sigma_{22}}{\sigma_T^+}\right)^2 + \left(\frac{\sigma_{12}}{\tau_A}\right)^2 = 1 \tag{4}$$

Experimental and analytical results are shown in figure 3.

In order to examine the problem related to failure criteria, S-N curve data are used instead of plane stress information. Thus,

$$F(\sigma_{ij}, R, N) = 1 \tag{5}$$

where R is the ratio of minimum and maximum amplitude in constant amplitude cycling and N is the number of cycles to failure (lifetime). Equation (5) represents a family of failure surfaces in stress space defined by the parameter N.



Tensile   Compressive

FIGURE 1. FIGURE MODES

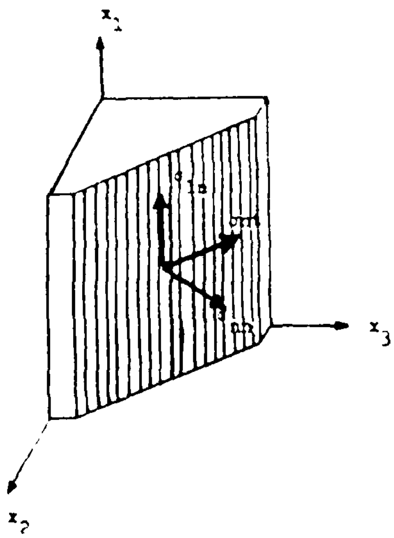


FIGURE 2. MATRIX MODES

Utilizing the transverse isotropy of the material with the quadratic approximation, Hashin has shown that for fully reversed cyclic loading, R = -1, the failure criteria are given by

$$\left(\frac{\sigma_{11}}{\sigma_A}\right)^2 + \left(\frac{\sigma_{12}}{\sigma_A}\right)^2 = 1 \qquad \text{Fiber Mode}$$

$$\left(\frac{\sigma_{22}}{\sigma_T}\right)^2 + \left(\frac{\sigma_{12}}{\sigma_A}\right)^2 = 1 \qquad \text{Matrix Mode} \tag{6}$$

The results found using the specimen in figure 4 agree reasonably well with theory even though the cyclic stress ratio is R = 0.1. See figures 5 and 6.

Hashin (reference 60) emphasizes that a fundamental problem concerning the engineering use of fiber composites is the determination of their resistance to combined states of cyclic stress. Analysis of fatigue failure based on the stresses obtained is not possible without failure criteria for three-dimensional information.

Since the damage which occurs during fatigue cycling is so complex, it is possible only to develop fatigue failure criteria for cyclic stress by using fatigue failure criteria for simple states of stress.

Hashin continued his investigation by considering the scatter problem. He assumed that the different lifetimes due to scatter for identical specimens are due to the differences in microstructure. Therefore, if a specimen could be reproduced exactly, it would exhibit no lifetime scatter. However, these specimens follow some type of deterministic failure criteria.

Assuming that the specimens all fail in the matrix mode and follow the equations developed previously

$$\left(\frac{\sigma_{22}}{\sigma_T}\right)^2 + \left(\frac{\sigma_{12}}{\sigma_A}\right)^2 = 1 \tag{7}$$

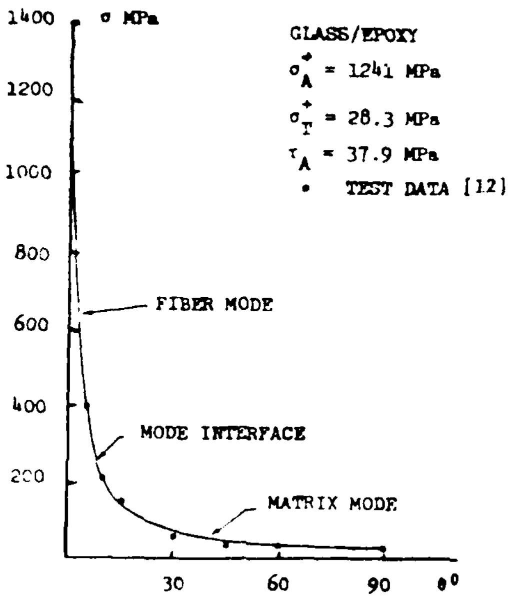


FIGURE 3. STATIC OFF-AXIS SPECIMEN DATA THEORY AND EXPERIMENT

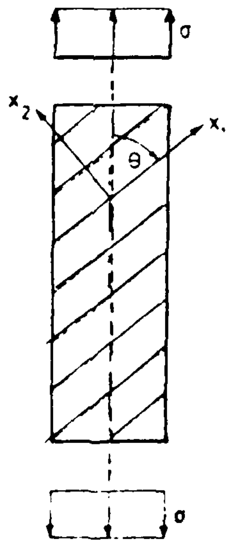


FIGURE 4. OFF-AXIS SPECIMEN

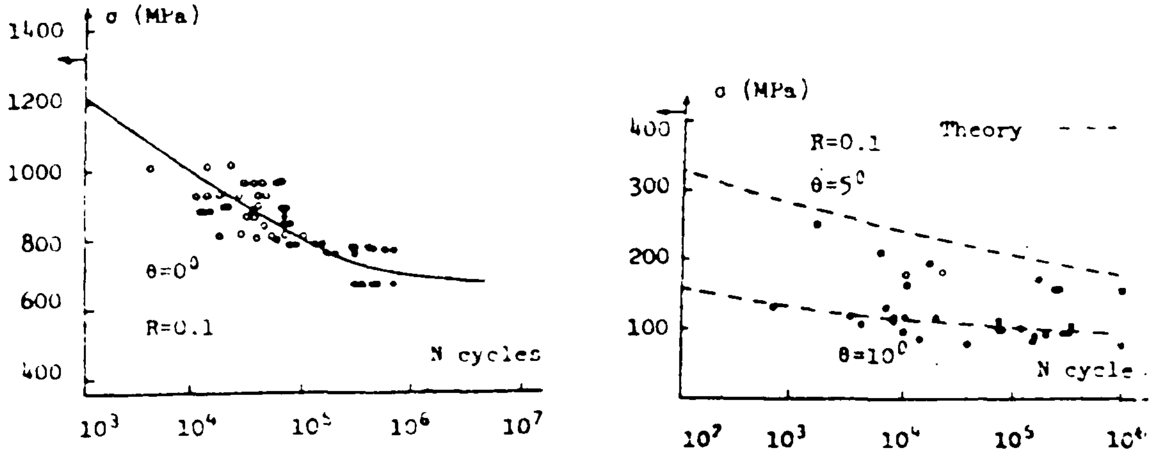


FIGURE 5. FATIGUE OFF-AXIS SPECIMEN DATA
THEORY AND EXPERIMENT
(EXPERIMENT NO. 1)

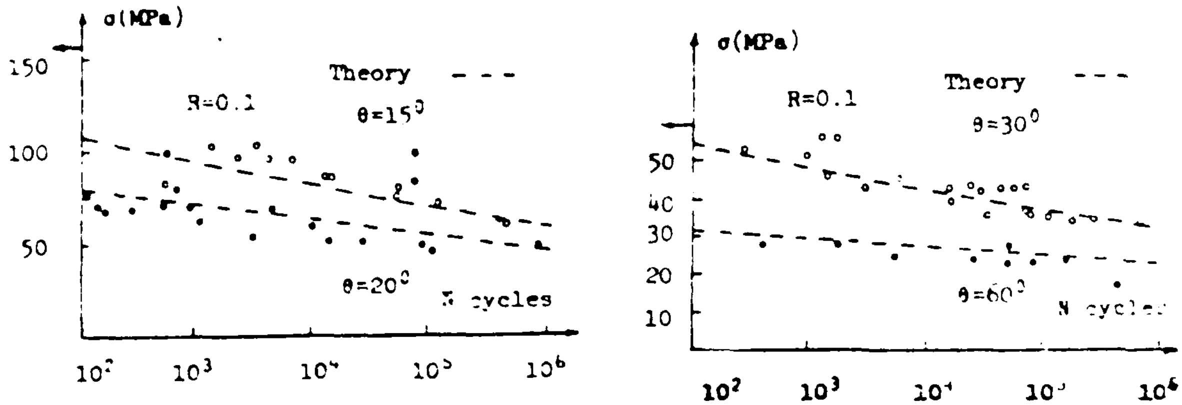


FIGURE 6. FATIGUE OFF-AXIS SPECIMEN DATA
THEORY AND EXPERIMENT
(EXPERIMENT NO. 2)

the following reasoning is suggested

$$\sigma_{22} = \sigma s_{22}$$
$$\sigma_{12} = \sigma s_{12} \qquad (8)$$

where s_{22} and s_{12} are nondimensional. Using S and N criteria,

$$\sigma_m(R, N) = \left[\left(\frac{\sigma_{22}}{\sigma_{Tm}}\right)^2 + \left(\frac{\sigma_{12}}{\tau_{Am}}\right)^2\right]^{-\frac{1}{2}} \qquad (9)$$

Using M specimens, the mean of V is

$$\langle\sigma\rangle(R, N) = \frac{1}{M}\sum_{m=1}^{M}\sigma_m\left[\sigma_{T_m}(R, N),\ \tau_{A_m}(R, N)\right] \qquad (10)$$

and the variance v is

$$v(R, N) = \frac{1}{M}\sum_{m=1}^{M}\left[\sigma_m - \langle\sigma\rangle\right]^2 \qquad (11)$$

The mean failure stresses are given by

$$\langle\sigma_{22}\rangle = \langle\sigma\rangle s_{22}$$
$$\langle\sigma_{12}\rangle = \langle\sigma\rangle s_{12} \qquad (12)$$

Hashin points out the fact that there are practical difficulties, since stress can be controlled by a testing machine during fatigue testing, but N cannot be controlled. Thus, in order to perform a "vertical" stress average (see figure 7), it is necessary to have a very large number of test data.

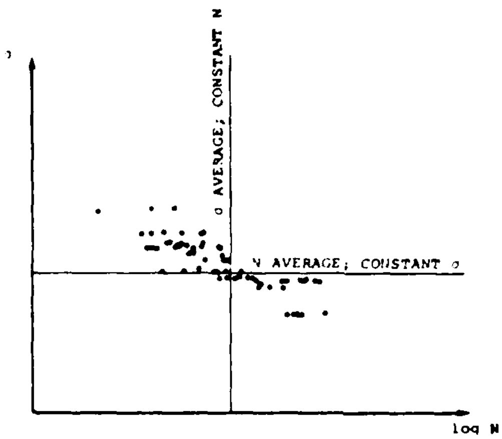


FIGURE 7. TEST DATA AVERAGING

A characterization (or characterizing) of a composite material in regard to fatigue is simply a description of characteristics or peculiar qualities. As it was mentioned earlier, Whitney (reference 24) developed a procedure that allows the generation of an S-N curve with some statistical value without resorting to an extremely large data base. He assumes a direct relationship between static strength distribution, residual strength distribution, and distribution of time-to-failure at a maximum stress level. This approach is called the "wearout" or "strength degradation" model. Whitney's model is compatible with the wearout model, but does not require any relationship between the fatigue life and residual strength.

Whitney (reference 24) used a fatigue characterization model used by Hahn and Kim (reference 61) which assumes a power law S-N curve and a two-parameter Weibull distribution to failure. These assumptions are manifested in the following equations

$$CNS^b = 1 \qquad (13)$$

$$R(N) + \exp\left[-\left(\frac{N}{N_o}\right)^{\alpha f}\right] \qquad (14)$$

where R(N) is the reliability of N (probability of survival), N_O is the characteristic time to failure, and α is the fatigue shape parameter.

While a plot of log S versus log N_O produces a straight line, it is more advantageous to construct an S-N curve. Thus,

$$S = K\left\{\left[-\ln R\ (N)\right]^{\frac{1}{\alpha_f b}}\right\} N^{-\frac{1}{b}} \qquad (15)$$

for any level of reliability.

In order to reduce the data, Whitney used a two-parameter Weibull distribution to fit the time-to-failure data at each stress range, obtained through a data pooling system, and determined b and k by plotting log S versus log N_O. $\hat{N}_{oi}$, the estimated life, is found from

$$\hat{N}_{oi} = \frac{1}{n_i} \sum_{j=1}^{n_i} \hat{N}_{ij}^{2} \frac{1}{2f_i} \qquad (16)$$

Yang and Du (reference 62) have investigated one of the important problems in the design of aircraft structures. This problem deals with the prediction of the fatigue behavior of composite laminates or joints subject to service loading spectra. Since the fatigue model is based on failure mechanisms, it is independent of stacking sequence. Herein lies its advantage over other models.

The two-parameter Weibull distribution is used to describe the pattern of the ultimate strength. Normalized ultimate strength is represented by

$$F_{R(o)}(x) = P\left[R(o) \leq\right] = 1 - \exp(-x^{\alpha}) ; \quad x \geq 0 \qquad (17)$$

where $F_{R(o)}(x)$ is the distribution function of R(o) and P indicates the probability of the ultimate strength occurring. At the end of the first cycle block (mission), the normalized residual strength is

$$R(1) = R(o) - \Phi\left[R(o)\right] \tag{18}$$

where [R(o)] is the reduction of the residual strength resulting from the application of one cycle block. Therefore, the normalized residual strength at the end of m cycle blocks is

$$R(m) = R(o) - m\Phi\left[R(o)\right] \tag{19}$$

The normalized absolute maximum stress in the cycle block is

$$\sigma_{max} = \max_{1 \leq i \leq L} \left|\sigma_i\right| \tag{20}$$

where it is assumed that fatigue failure occurs when the applied stress exceeds the residual strength. Now

$$y = \left(-\ln\left[1 - F_M(m)\right]^{\frac{1}{\sigma}}\right) \tag{21}$$

and

$$m = \frac{y - \sigma_{max}}{\Phi(y)} \tag{22}$$

The distribution function of the fatigue life is

$$F_M(m) = P\left[M \leq m\right] = P\left[R(m) \leq \sigma_{max}\right] = F_{R(m)}\ \sigma_{max} \tag{23}$$

Yang and Jones (reference 63) used the three-parameter fatigue and residual strength degradation model for unnotched composite laminates to describe the effect of load sequence on the statistical distribution of the fatigue life and the residual strength under n - stress levels of cyclic loading. The following equation resulted

$$R^c(n_1) = R^c(o) - \beta^c K S^b n \qquad (24)$$

where $R(n_1)$ and $R(n_0)$ are the residual strengths at n_1 and n_0 cycles ($n_1 > n_0$) respectively, β is the scale parameter of the ultimate strength, b, c, and k are three parameters to be determined by tests, S is the stress range, $\tau_{max}-\tau_{min}$, and R is the stress ratio. Letting $n_0 = 0$ and $n_1 = 1$ where R(o) is the ultimate strength.

It is assumed that the ultimate strength is a statistical variable and follows the two-parameter Weibull distribution.

$$F_{R(o)}(x) = P\left[R(o) \leq x\right] = 1 - \left[\exp -(x)\right] \qquad (25)$$

After applying high-low and low-high constant stress amplitude load sequences, they arrive at the following equation for the fatigue life,

$$\begin{aligned} F_{N_{12}}(n) &= P\left[N_{12} \leq n\right] \\ &= P\left[R^c(o) - \sigma_{2max} - \beta^c KS_1^b \leq N\beta^c KS_2^b\right] \qquad (26) \\ &= 1 - \exp\left\{-\frac{n}{N_2^*} + \frac{n}{N_1^*} + \frac{\sigma_{2\,max}}{\beta}^{c\;\frac{\alpha}{c}}\right\} \end{aligned}$$

which exhibits the characteristics of a three-parameter Weibull distribution.

Using a probability expression that describes the fatigue failure at the nth load cycle

$$p_n = P\left[R(n-1) \le \sigma_{nmax}\right]$$

$$= 1 - \exp\left\{-\left[\left(\frac{\sigma_{nmax}}{\beta}\right)^c + \sum_{i=1}^{n-1} \frac{1}{N_i^*}\right]^{\frac{\alpha}{c}}\right\} \qquad (27)$$

Thus, the distribution of the fatigue life is

$$F_N(n) = P\left[N \le n\right]$$

$$= 1 - \exp\left\{-\left[\left(\frac{\sigma_{nmax}}{\beta}\right)^c + \sum_{i=1}^{n-1} \frac{1}{N_i^*}\right]^{\frac{\alpha}{c}}\right\} \qquad (28)$$

Finally, Yang and Jones establish that the statistical distributions of both the fatigue life and the residual strength do not depend on the load sequence before the nth load cycles.

Coupon specimens of 5208/T3000 graphite/epoxy $[+45°]_{2s}$ laminates were tested using information from research done by Rosen (reference 64) and Hahn (reference 65). Results illustrate the correlation between the theoretical and measured fatigue life distributions. It was shown that the model is not a linear cumulative damage model.

CORRELATION OF EQUATIONS WITH COMPONENTS.

Demuts, Whitehead, and Deo (reference 66) conducted experiments on carbon/epoxy coupons and built-up panels undergoing uniaxial loading. Damage tolerance to processing and normal service were assessed during these experiments. Using data derived from studies of previous researchers and tests performed in this study, they correlated the results found for coupons with those of two-bay built-up panels found in multispar and multirib wing skin designs.

The most severe relative strength loss was due to low velocity impact with a blunt semispherical-shaped impactor. The size of this impactor ranged from 1.27 centimeters to 2.54 centimeters in diameter. The impactor velocity was 46 m/sec. Damaged areas were measured, and it was found that a damage line of 3.7 centimeters corresponded to a strength loss of 58 percent, and a damage line of 4.9 centimeters corresponded to a strength loss of 73 percent. They concluded that a structure which has not been designed adequately for damage tolerance may fail without exhibiting any visible signs of damage. See figures 8 through 11.

The M-R panels have higher static strength than the M-S panels. It was found that damage grew in both panels when the constant amplitude compression-compression (R = 10) fatigue load severity reached 65 percent of the damaged static strength. It was pointed out that damage growth is not well characterized.

Williams, O'Brien, and Chapman (reference 67) emphasize that an important consideration in attaining the potential structural efficiency improvements with resin matrix composite structures is the need to improve their resistance to impact damage which may occur in normal service, and to improve resistance to delamination which could result from unanticipated out-of-plane loads. This has resulted in manufacturers directing their attention toward developing materials with tougher resin matrices. Further, toughness is defined as the ability to deform elastically under interlaminar shear and peel stresses without undergoing brittle fracture, which many of the present resin matrices are experiencing.

In order to meet this new challenge of toughness of new materials and additional requirements, the NASA Aircraft Energy Efficiency (ACEE) Project Office and their industrial contractors have identified and selected a set of "standard tests" which are now used by all the ACEE contractors and researchers at Langley Research Center. The five tests include interlaminar fracture (edge delamination tension and double beam cantilever test), notch sensitivity (open-hole tension and compression test), and the effect of impact damage on compression strength. NASA specifications for standard tests were followed. See figure 12.

Williams and Rhodes (reference 68) have developed a tension test to be used to measure interlaminar fracture toughness of composites using tough matrix resins. The modulus, E_{lam}, and the nominal strain at the beginning of edge delamination are measured while the tension test of an 11-ply or 8-ply laminate is tested.

The strain energy release rates, G, are solved in a closed form equation for evidence of edge delamination growth (reference 69). The E^* term is the modulus of the laminate if the 0/90 interface is completely delaminated, and G_c is a measure of the interlaminar fracture toughness.

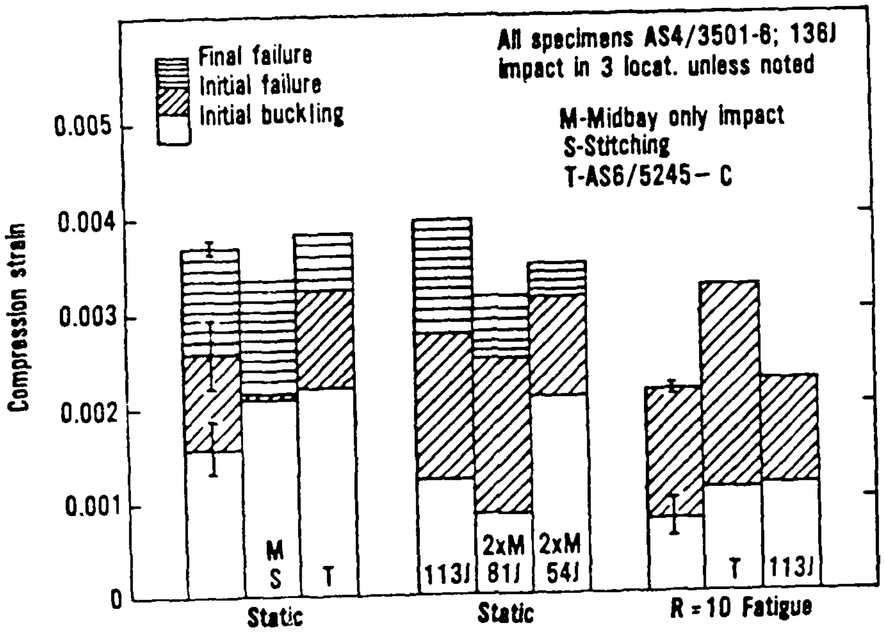


FIGURE 8. THREE-SPAR PANELS - IMPACT DAMAGE STRENGTH

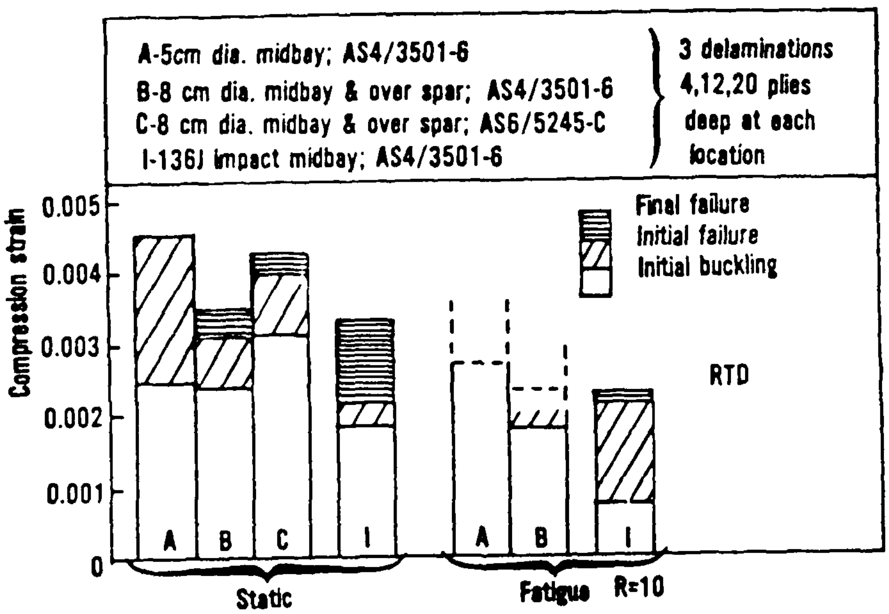


FIGURE 9. THREE-SPAR PANELS - DELAMINATION STRENGTH

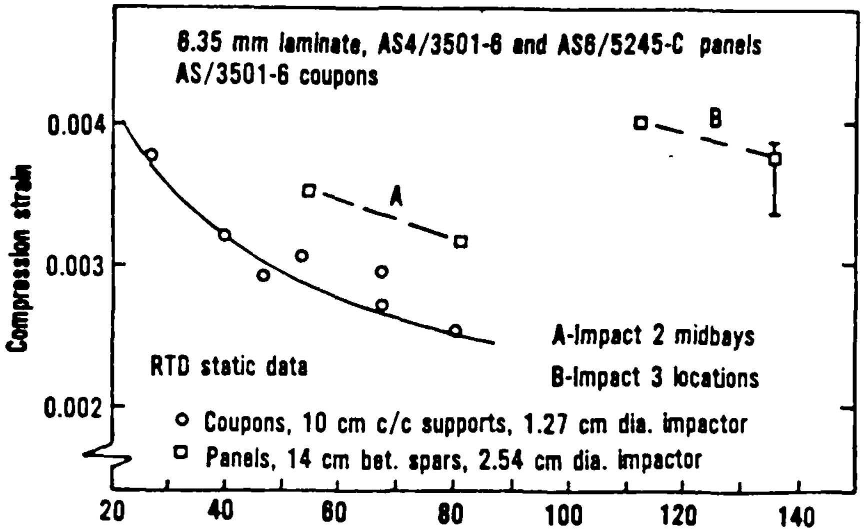


FIGURE 10. THREE-SPAR PANELS - SPECIMEN COMPLEXITY EFFECT

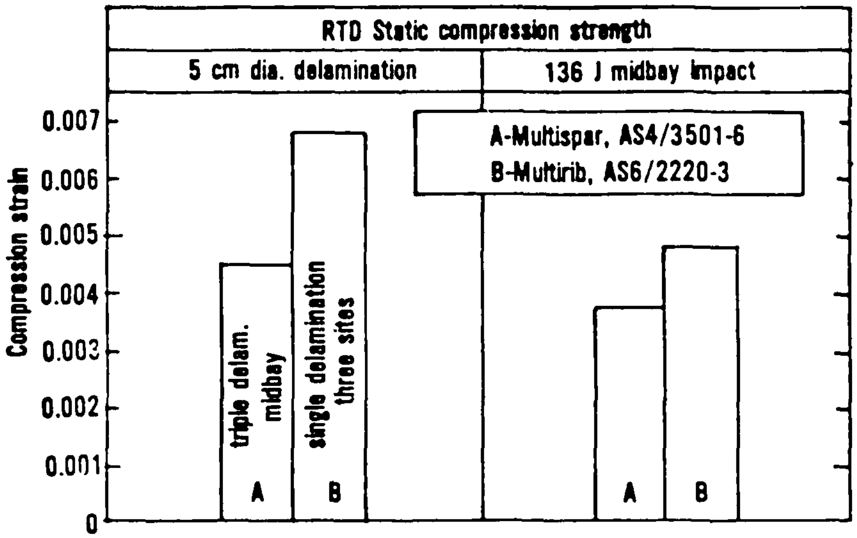


FIGURE 11. PANEL DESIGN COMPARISON - STATIC STRENGTH

Regarding damage tolerance, Williams, Anderson, Rhodes, Starnes, and Stroud (reference 70); and Carlile and Leach (reference 71) are responsible for the aforementioned test procedures. Using the low-mass/high velocity test methods appears to cause the most reduction in strength.

Haftka, Starnes, and Nair (reference 72) studied global damage tolerance and the mass penalty associated with improving the global damage tolerance of optimized aircraft wing structures. In order to establish damage tolerance criteria, structures would be required to carry a percentage of the design load when a major structural member is destroyed.

Using three examples, they show that the mass of the damage tolerance design depends on the structural redundancy and the percentage of load being carried in the damage configuration. See figures 13, 14, and 15.

The buckling load is,

$$N = \frac{\pi^2 Et}{3(1 - \nu^2)L^2} \tag{29}$$

where E is the effective longitudinal laminate modulus and ν is the laminate Poisson's ratio. After damage, the plate can carry a fraction, f, of the original undamaged buckling load. Letting,

$$\frac{t_1}{t_2} = \beta \tag{30}$$

the residual strength is calculated based on the assumption that the effective modulus and Poisson's ratio remain the same, and it is assumed that there is no postbuckling stiffness. For a residual strength of (fxN)

$$r = \frac{\left(1 + \beta^3\right)}{\left(1 + \beta\right)^3} \tag{31}$$

The residual strength has its lowest value when ($\beta = 1$) and ($f = 0.25$). If

$$r(x)_0 = \frac{(1 + \beta^3)}{(1 + \beta)^3} \tag{32}$$

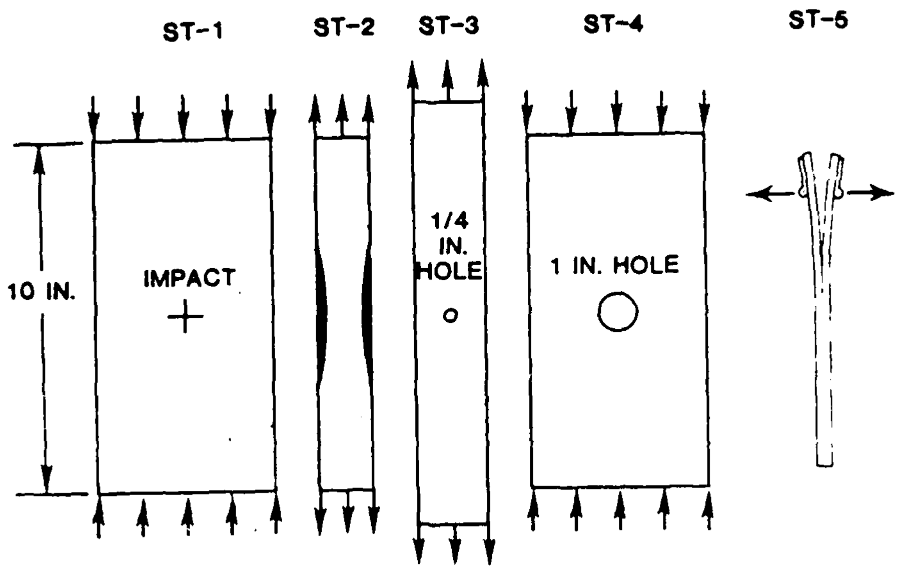


FIGURE 12. STANDARD TESTS FOR TOUGHENED COMPOSITES

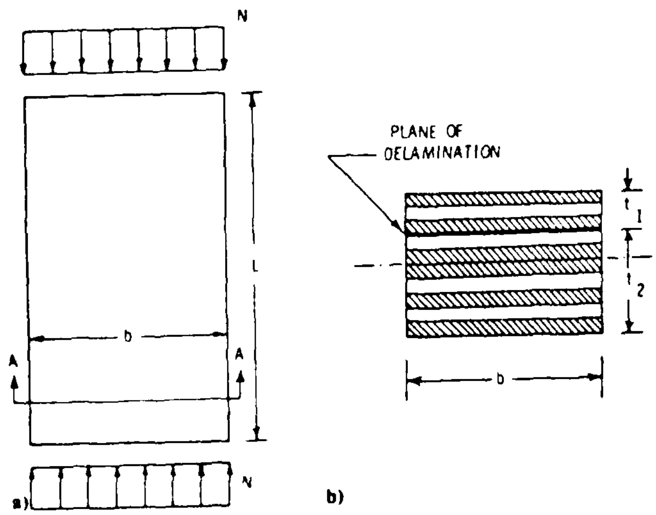


FIGURE 13. GEOMETRY AND LOADING OF RECTANGULAR PLATE DELAMINATION. EXAMPLE: a) GEOMETRY AND LOADING; b) VIEW A-A SHOWING LAMINATE CROSS-SECTION

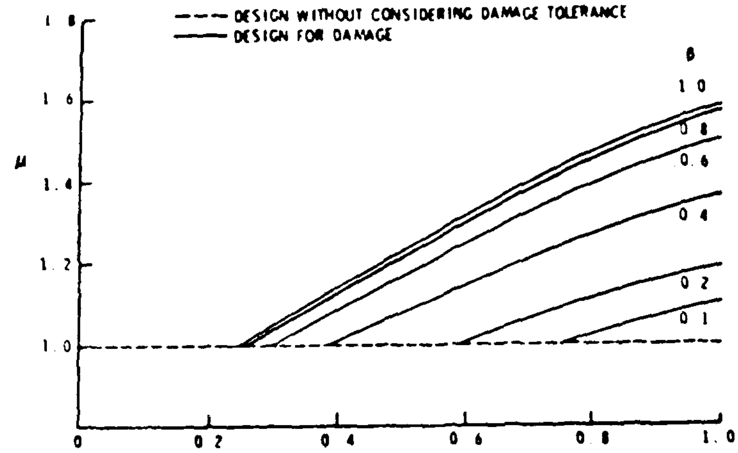


FIGURE 14. VARIATION OF NORMALIZED STRUCTURAL MASS WITH REQUIRED RESIDUAL LOAD FACTOR f FOR DIFFERENT VALUES OF = t_1/t_2

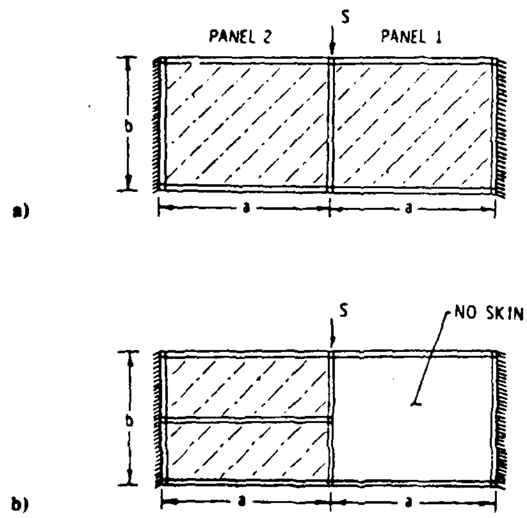


FIGURE 15. TWO-BAY DEEP BEAM: a) ORIGINAL CONCEPT
b) ALTERNATE CONCEPT

($f = 1$) and ($\beta = 1$), and ($\mu = 1.59$) the mass penalty for damage tolerance is 59 percent. Thus, there is a great deal still not known about fatigue of complex composites. In particular, it was found that damage to the tension cover-skin panel reduces the strength of the wing more than damage to the compression cover-skin panel.

Poe (reference 73) investigated the damage tolerance of bonded composite stringers loaded in tension. Tensile failure tests on 50 graphite/epoxy composite panels were made with two sheet layups and several stringer configurations. Slits were cut in the middle of the panels. See figure 16 for the configuration of stiffened panels. Figures 17 and 18 show test results for panels with ($\mu = 0.7$) and ($\mu = 0.5$). Figure 19 shows a shear-lag analysis where the stiffness is given by,

$$(Et)_1 = (Et)_{sh} + (Et)_{st} \tag{33}$$

E_{sh} and E_{st} are the sheet and stringer Young's moduli. For a large effective crack width,

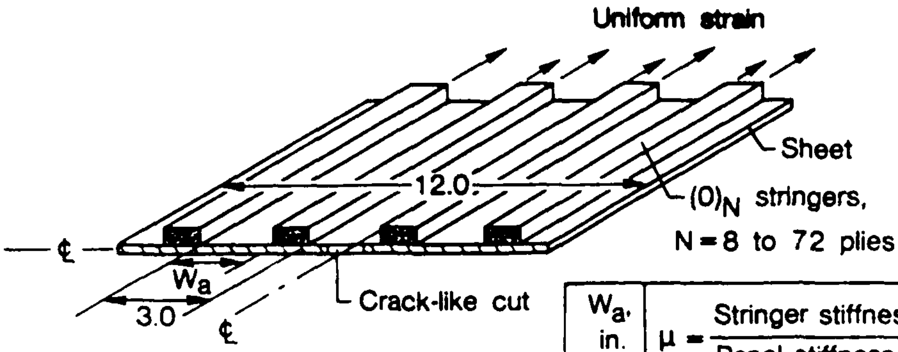
$$SCF \propto \sqrt{\frac{Wa}{\left[\dfrac{Wa}{(1+\alpha)}\right]}} \tag{34}$$

Figures 20, 21, 22 and 23 show that stress intensity factors can be synthesized. However, substituting,

$$K_Q = \frac{Q_c E_x}{\xi} \tag{35}$$

into

$$\frac{\varepsilon_{tu}}{\varepsilon_c} = \sqrt{\frac{1 + \pi Wa (Et)_{sh} \, {}_{tu}^{2} \, E_{sh}}{(Et)_1 \, K_Q}} \tag{36}$$



Material: T300/5208

Sheet layups: $(45/0/-45/90)_{2S}$ and $(45/0/-45/0)_{2S}$

W_a, in.	$\mu = \frac{\text{Stringer stiffness}}{\text{Panel stiffness}}$
2.0	0.3
.5 2.0 2.5	.5
2.0	.7

FIGURE 16. CONFIGURATION OF STIFFENED PANELS

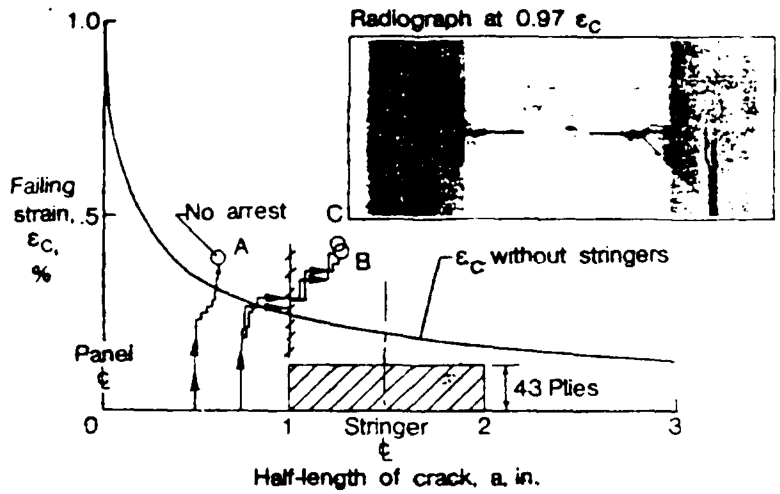


FIGURE 17. FAILING STRAIN VERSUS CRACK-TIP POSITION

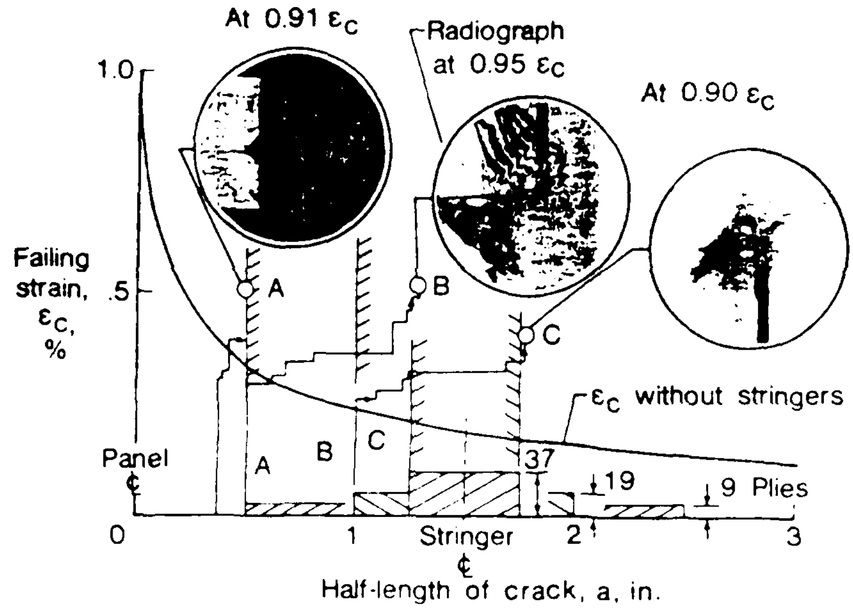


FIGURE 18. FAILING STRAIN VERSUS CRACK-TIP POSITION

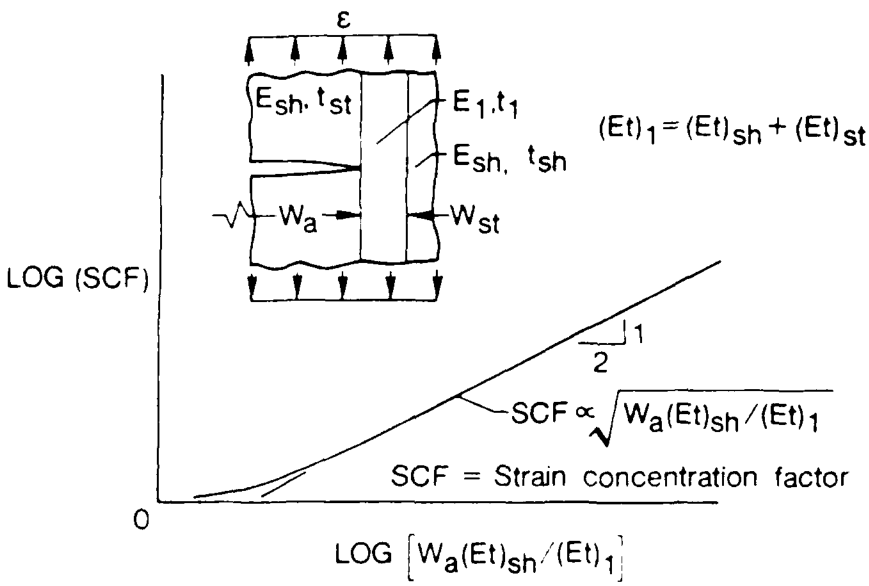


FIGURE 19. RESULTS FROM SHEAR LAG ANALYSIS

Without stringers,

$$K_Q = S_{sh}\sqrt{\pi a + K_Q^2/F_{tu}^2} \qquad (1)$$

For uniaxial stress ($S_{sh} = E_{sh}\,\varepsilon_c$ and $F_{tu} = E_{sh}\,\varepsilon_{tu}$),

$$\frac{\varepsilon_{tu}}{\varepsilon_c} = \sqrt{1 + \frac{\pi a\,\varepsilon_{tu}^2 E_{sh}^2}{K_Q^2}} \qquad (2)$$

Where

$$SCF = \varepsilon_{tu}/\varepsilon_c$$

With stringers, replace a by $1/2\ W_a\,(Et)_{sh}/(Et)_1$

$$\frac{\varepsilon_{tu}}{\varepsilon_c} = \sqrt{1 + \frac{\pi W_a\,(Et)_{sh}\,\varepsilon_{tu}^2 E_{sh}^2}{2\,(Et)_1\,K_Q^2}} \qquad (3)$$

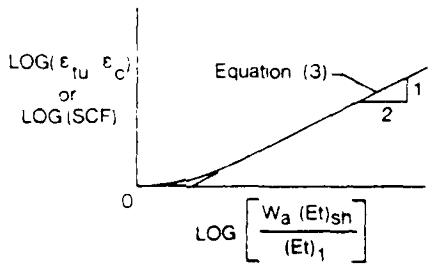


FIGURE 20. SYNTHESIZED STRESS INTENSITY FACTOR

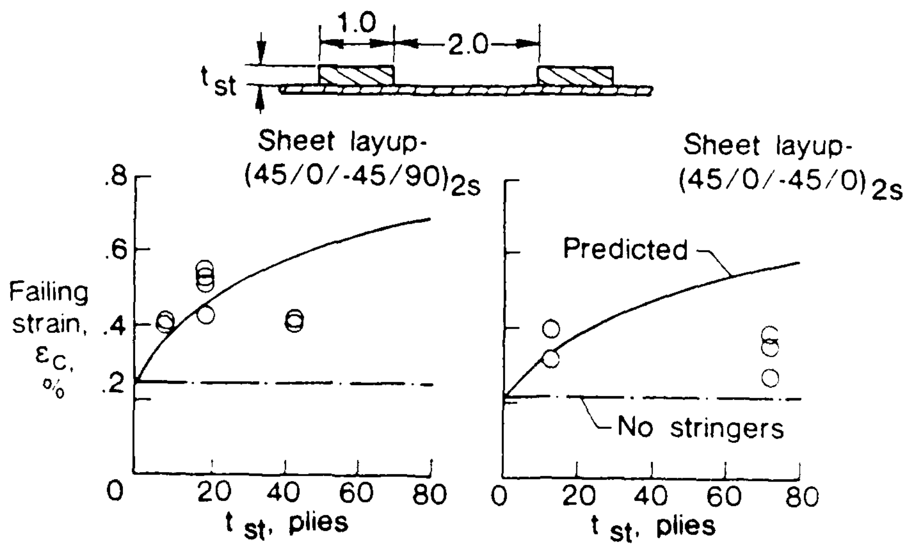


FIGURE 21. FAILING STRAIN VERSUS STRINGER THICKNESS

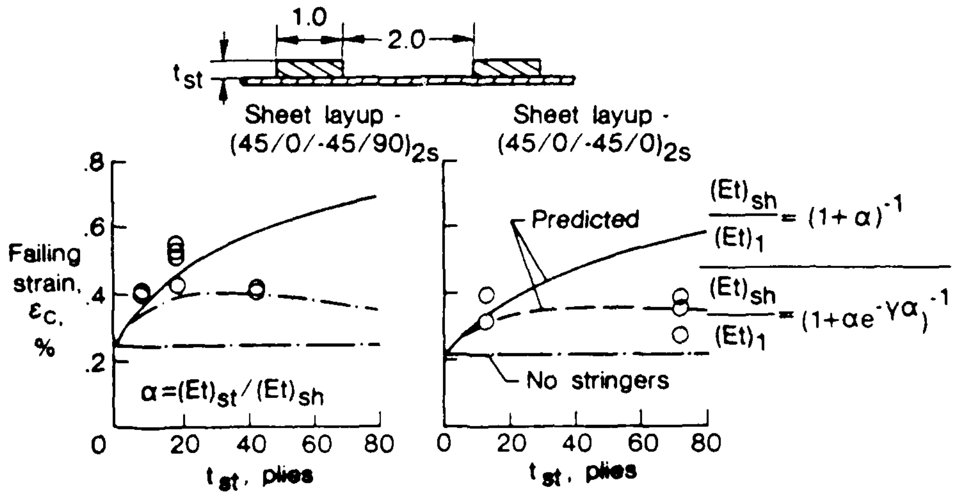


FIGURE 22. FAILING STRAIN VERSUS STRINGER THICKNESS

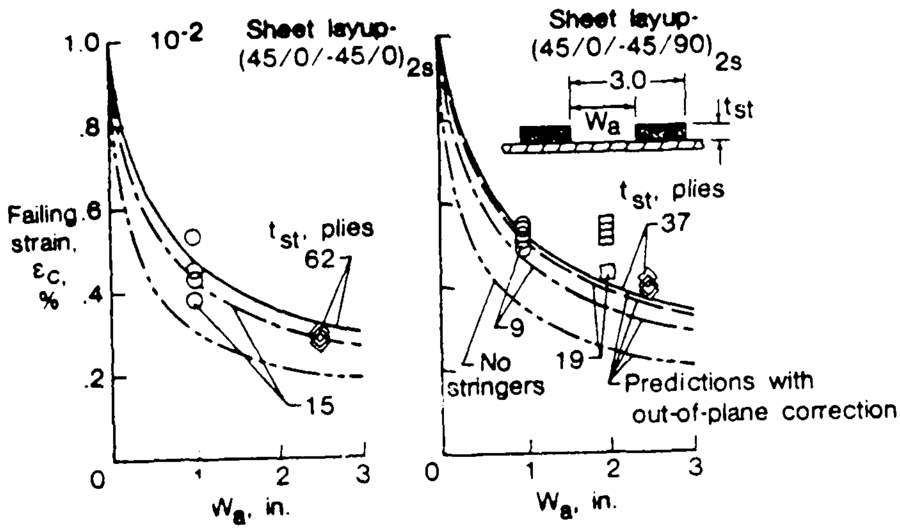


FIGURE 23. FAILING STRAIN VERSUS STRINGER SPACING

ingle design equation is introduced for stiffness panels with any sheet layups and made of any material. The equation is shown as a single design curve in figure 24.

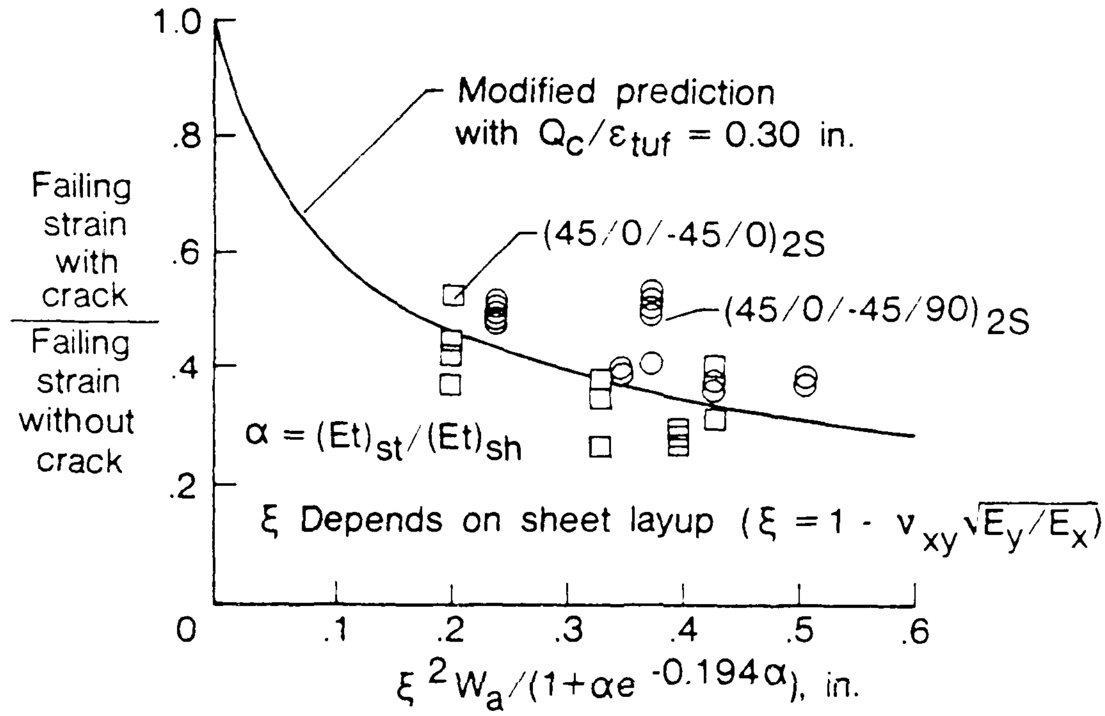


FIGURE 24. DESIGN CURVE FOR STIFFENED PANELS

CORRELATION OF BASIC EQUATIONS OF IMPORTANCE WITH RESULTS FOUND FROM TESTS BY AIRCRAFT MANUFACTURERS' COMPONENT

Commensurate with the correlation of basic equations of importance, results found from tests by aircraft manufacturers performed on components are as follows:

1. In July 1983, Ansell (reference 74), head of research at Rensselaer Polytechnic Institute's (RPI) Composite Aircraft Program Component (CACOMP), studied the fatigue load transfer that took place between the connecting lugs at the ends of the structure and the portions of the well-distributed structure. The drag strut of the Lockheed L1011 is an example of the primary structure made of graphite/epoxy. In the concluding remark, RPI demonstrated a load/weight graphite efficiency of 425 percent greater than steel.

2. Rotem (reference 75) of Advanced Research and Applications Corp. in conjunction with Ames Research Center, presented a detailed description of fatigue functions - the influence of temperature in tension-tension fatigue behavior of graphite/epoxy. Lockheed-fabricated specimens were highly representative of the components which were tested.

3. In April 1981, Ramkumar (reference 76) of Northrop Corp. reported in his paper the compression fatigue behavior of graphite/epoxy laminates in the presence of imbedded delaminations. Three different stacking sequences of a quasi-isotropic layup (0/45/90-45, 45/90-45/0, and 90/45/0-45) and 64-ply thick specimens were provided by Lockheed as part of a component. His report provided very detailed results of compression and fatigue-life tests.

4. Approximately three months earlier, Lieblein (reference 77) conducted a survey of the long-term fatigue strength properties of fiberglass-reinforced plastic structures. Included in the survey were data from aircraft radomes with up to 19 years of service, such as the fiberglass laminate rotodome of the E2A and the filament woven fiberglass nose radome of the A6, both manufactured by Grunman Aerospace Corp.

5. In the same period, RPI presented the CACOMP in conjunction with the Boeing Company. RPI's research team, headed by Ansell (reference 78), provided a detailed report of fatigue testing and analytical work on the main spar/rib for the Boeing 727 elevator.

6. In 1980, an investigation was conducted by Rhodes (reference 79), Structural Mechanics Branch of NASA Langley Research Center, to study the damage tolerance of composite compression panels using graphite/epoxy specimens. Damage due to impact by a 1.27 centimeter diameter spherical projectile was representative of wing-skin panels. The results indicated that substantial improvements in the damage tolerance of graphite/epoxy structures can be achieved through the proper combination of materials and structural design.

7. In the proceedings of the Japan - U.S. Conference in Tokyo, 1981, Tada, Ishikawa, and Nakai (reference 80) conducted tests to perform weight reduction in the aircraft structure of the quiet STOL research aircraft. Finite element analysis was used to obtain results of load-displacement and load-strain relations for the carbon fiberglass reinforced plastics of the rib/spar models. Stiffness moduli of a composite plate were also reported in their paper.

8. From the Technical R&D Institute of the Japan Defense Agency, Yamauchi and Mogami (reference 81) explained the application and development of the advanced composite ground spoiler for C-1 medium transport aircraft. Included in the report was an explanation for the application of graphite/epoxy materials with high strength. This stiffness ratio was representative of the weight characteristics of the C-1 ground spoiler. In the same conference, Yamauchi and Mogami (reference 82) also presented the

design, fabrication and development of graphite/epoxy rudders for flight tests of the T-2 jet trainer. The primary objective was to develop a flight certifiable test program in order to evaluate composite structures in an actual flight environment. The paper also covered environmental effects and compared the behavior of skin plies undergoing orientation, durability and vibration tests.

9. Finally, Takagi and Idei (reference 83) conducted a structural test program for Fuji Heavy Industries and included a full-scale test (rigidity and fatigue strain tests) for the T-2 military jet trainer. The paper also describes the development status of the composite vertical stabilizer for T-2, the graphite/epoxy helicopter tail rotor, the graphite/epoxy T-2 rudder, and the Kevlar-graphite/epoxy 767 wing/body fairing and main landing gear.

The matrix (table 1) which identifies fatigue composite material usage in commercial/civil aircraft is based primarily on information taken from Jane's All The World's Aircraft. Also, where appropriate, and particularly in regard to presenting the latest information possible, the aforementioned information has been supplemented with data from other sources.

For this report, in concert with FAA, aircraft manufacturers, NASA, and Schwartz (reference 84), primary structures are defined as horizontal stabilizers and vertical firms. Considering developmental work based on extensive, recent studies (reference 85), vertical stabilizers and wings were included as primary structures. Also commensurate with this rationale, secondary structures include edges, spoilers, rudders, cambers, fairings, and control surfaces.

Information included in the matrix represents the digestion and cullation of data from at least 69 sources within Jane's All The World's Aircraft and other related sources. While commerical/civil aircraft information is included in source numbers 1 through 43, a limited amount of military-related comparison information, while not required, has been included in source numbers 44 through 69 in order to enhance the report appropriately.

In order to highlight those aircraft with high amounts of composite usage, in concert with the study of Composite Material Fatigue/Damage Tolerance, an asterick ("*"-high amount, "**"-significant amount) has been assigned to the corresponding aircraft manufacturer. The Reference Number serves as project indentification for the materials extracted from Jane's All the World's Aircraft, 1975 to 1984.

TABLE 1. MATRIX FOR AIRCRAFT FATIGUE COMPOSITE MATERIAL USAGE

Commercial/Civil

Ref No.	Manufacturer [Type,Year]	COMPOSITION					APPLICATION									DESIGN	
							Wing					Tail					
							P	S	S	S		P	S				
		Graphite	Glass	Kevlar	Carbon	Boron	Structure	Edges/Spoiler	Rudder	Camber	Fuselage	Structure	Control Surf	Rotor Blades	Landing Gear	Unidirectional	Non-Woven
1.	Avtek [400. 6/83]	X		X			X						X				
2.	Boeing [737, 4/81]	X						X	X				X				
3.	Boeing [757, 2/82]	X						X									
4.	Boeing [767, 8/82]	X						X									
5.	Boeing Verto [234, 1983]		X												X		
6.	Cessna [152, 1983]		X					X									
7.	Cessna [Stationair, 1983]		X							X							
8.	Cessna [Centurion, 1983]		X							X							
9.	Cessna [Citation, 1982]	X			X			X									
**10.	Composite Aircraft [Eagle, 80]	X	X				X					X					
11.	Hillman [360, 1981]		X											X			
12.	Hughes [500, 1981]		X											X			
**13.	Lear Fan LTD. [2100, 1982]	X		X			X	X									
14.	Lockheed [L-1011, 1983]	X	X									X	X				
15.	McDonnell Douglas [DC-9, 1978]		X		X			X									
16.	McDonnell Douglas [MD80, 83]		X	X	X		X	X			X	X					
17.	McDonnell Douglas [MD100, 83]								X				X				
*18.	Gyroflug [Speed Canard, 83]		X		X		X										
19.	Airtech [CN 235, 83]	X	X					X									
*20.	SAAB-Fairchild [340, 82]		X	X			X	X			X		X				
21.	Piper [PA32301, 1979]		X				X										
22.	Piper [PA31325, 1983]		X					X									
23.	Piper [PA31350, 1981]		X					X									
24.	Mike Smith Aero [XD99, 1981]	X	X	X							X		X				
25.	Varga [2150A, 1981]		X					X					X				
*26.	Ames Indust Corp [AD-1, 79]		X								X		X		X		
*27.	Omnionics [Dolphinair, 82]		X	X			X										
28.	Spitfire [Mark II, 1979]		X												X		
29.	Swearingen [SA226TC, 80]		X								X						
30.	Airbus Industrie [A300, 82]				X			X									
31.	Bellanca [Skyrocket, 82]		X		X			X									
**32.	Rutan [Erizzly, 1982]		X		X		X										
*33.	Schape [5350, 1981]		X		X		X				X						
34.	Mudry [Cap21, 1980]		X					X									
35.	Farrington [18-A, 1974]		X											X			
36.	Turner [T-40A, 1974]		X					X			X						
37.	Bede [BD-7, 1975]		X					X							X		
38.	Miller [JM-2, 1974]		X					X			X				X		
**39.	Rutan [Variviggen, 1972]		X				X								X		
**40.	Rutan [Varieze, 1974]		X				X				X				X	X	X
41.	Van [RV-3, 1976]		X					X					X		X	X	X
42.	McDonnell Douglas [DC-10, 83]			X				X				X	X				
**43.	Beechcraft [Starship]	X		X			X	X	X	X	X	X	X	X	X	X	X

P = Primary Structure

S = Secondary Structure

TABLE 1. (CONTINUED)

Military		COMPOSITION					APPLICATION									DESIGN	
							Wing					Tail					
							P	S	S	S		P	S				
Ref No.	Manufacturer [Type, Year]	Graphite	Glass	Kevlar	Carbon	Boron	Structure	Edges/Spoiler	Rudder	Camber	Fuselage	Structure	Control Surf	Rotor Blades	Landing Gear	Unidirectional	Non-Woven
44.	Boeing [737, 4/81]	X						X					X				
45.	Boeing Vertol [107, 1981]		X											X			
46.	Boeing Vertol [414, 1980]		X											X			
47.	Boeing Vertol [360, 1984]		X	X											X		
**48.	Composite [Eagle, 1980]	X	X				X					X				X	X
49.	Fairchild Republic [NGT, 82]			X				X									
50.	Gen. Dynamics [F16-B, 83]		X									X	X				
51.	Gruman [Hawkeye, 1981]		X									X					
52.	Gruman [F-14, 1981]				X								X				
53.	Gruman [FSW, 1982]	X					X	X									
54.	Gulfstream [Gulfstream IV, 82]				X				X								
55.	Kaman [Seasprite, 1984]		X											X			
56.	McDonnell Douglas [F18A, 82]		X								X						
57.	Sepecat [Jacuar, 1982]				X	X	X										
58.	Northrop [F-20, 1982]	X											X				
59.	Northrop [F-18L, 1982]	X						X			X						
*60.	Rockwell [B-1B, 1982]	X						X									
* 61.	Sikorsky [CH53E, 1982]		X											X			
* 62.	Sikorsky [UH-60A, 1982]	X	X	X							X			X			
* 63.	Sikorsky [S-76, 1982]	X	X	X							X			X	X		
64.	Bell [214ST, 1980]		X											X		X	X
65.	Cessna [337, 1980]		X							X							
66.	NGEA (WGER) [ALPHA Jet, 82]		X						X				X				
* 67.	McDonnell Douglas B/AE																
	[AV-8B, 81]				X	X					X	X					
68.	Panavia [Tornados, 1981]		X								X						
69.	Schapel [SA981, 1980]		X	X	X		X				X						

P = Primary Structure
S = Secondary Structure

NON-DESTRUCTIVE TESTING METHODS OF FIBROUS COMPOSITE MATERIALS

With respect to prospective non-destructive testing methods of fibrous composite materials, there are six presently recognized approaches:

a. Video Thermography Technique

b. Radiography

c. Surface Temperature Measuring Technique

d. Acoustic Emission Monitoring

e. Ultrasonic

f. Holography

VIDEO THERMOGRAPHY TECHNIQUE.

Real-time video-thermography can be used to investigate initiation and progression of subsurface damage caused by fatigue. This technique differs from most others because the materials are subjected to some steady-state mechanical energy, such as fatigue loads or low amplitude vibration, that activates heat sources near the damaged regions.

Experimental observations are discussed for a variety of composite materials including boron/aluminum, boron/epoxy, and graphite/epoxy by Henneke, Reifsnider, and Stinchcomb (reference 86).

RADIOGRAPHY.

Radiography includes a number of different techniques (X-ray diffraction, Gamma ray, Penetrant, etc.) but they are all basically alike in that a penetrating beam of radiation passes through an object. As it does, different sections of the object, as well as discontinuities, absorb varying amounts of radiation so that the intensity of the beam varies as it emerges from the object.

Olley (reference 87), using low frequency X-radiography, has detected forms of fatigue damage in foamed PVC/fiberglass-reinforced plastic composite panels.

In composites, radiography is used to determine fiber alignment, intimacy of contacts in bonded areas, defects in sandwich constructions, and in reviewing core damages including voids, porosity, fracture, damaged filaments, delaminations and contaminations.

O'Brien (reference 88), Yeung, Stinchcomb, and Reifsnider (reference 89), and Daniel, Schramm and Liber (reference 90) demonstrated the application of radiology to detect delamination and damage propagation in graphite laminates.

SURFACE TEMPERATURE MEASURING TECHNIQUE.

Surface temperature monitoring by thermocouples and temperature sensitive strips/coatings as applied to glass/epoxy laminates (-45 +45) was demonstrated by Nevadunsky et al. (reference 91). The purpose of this investigation was to detect early non-destructive inspection techniques.

ACOUSTIC EMISSION MONITORING (AEM).

This technique involves placing a series of piezoelectric transducers about the specimen, applying a load, and "listening" for slippage and debonding. Several studies have demonstrated the feasibility of acoustic emission for inspecting graphite, boron, and fiberglass parts by Weghreter and Horak (reference 92), Laroche and Bunsell (reference 93) and Kim (reference 94).

ULTRASONIC.

Ultrasonics, like radiography, includes a number of different techniques. In ultrasonic inspection, a beam of ultrasonic energy is directed into a specimen, and the energy transmitted through it is indicated. Yeung, Stinchcomb and Reifsnider (reference 89) applied this technique for the case of characterization of constraint effects on flaw growth. In another reported experiment, Daniel, Schramm and Liber (reference 90) also applied ultrasonic monitoring of flaw growth in graphite/epoxy laminates under fatigue loading.

The four ultrasonic methods used in composite testing are: (1) pulse echo, used to inspect fiberglass-to-fiberglass bonds and delamination in fiberglass laminates; (2) pulse echo reflection plate, used to inspect delamination in thin fiberglass or boron laminates; (3) through-transmission, used to inspect thick fiberglass laminates; and (4) resonant frequency, used to detect fiberglass-to-fiberglass bonds where the exposed layer is not too thick.

In order to direct the sound wave through the test material, it usually requires a liquid contact or sometimes liquid immersion of the part. Therefore, it is necessary to provide a pair of transducers on each side of the structure to be tested.

HOLOGRAPHY.

Holography is an optical technique based on the optical interference produced by superposition of coherent light waves reflected from the object under consideration (object beam) and those of a coherent reference beam. A laser is an ideal source of coherent monochromatic light.

One of the most important applications of holography is the measurement of small surface displacements in a body produced by mechanical or thermal loadings. Such applications were discussed by Rowlands and Stone (reference 95).

CRITIQUE OF NON-DESTRUCTIVE TESTING METHODS.

In regard to the aforementioned approaches, no one particular non-destructive technique can be used with certainty for all configurations. Test methods must be selected and tailored to each item. The geometry of the parts must also be taken into account when determining the most appropriate test media.

For inspection of damage after dynamic fatigue loading, ultrasonic techniques should be included among the most useful methods. However, to direct the sound wave through the test material usually requires a liquid contact. Ultrasonics is a valuable inspection means for smooth-surfaced fine-grained materials oriented in a particular scan plane. Generally, small voids cannot be detected.

Radiography is probably one of the oldest and most commonly used techniques, but it requires special precautions to avoid hazards from radiation. Films are relatively expensive, and processing can require considerable time.

Machine noise has long posed a problem in acoustic emission monitoring. This requires the use of noise insulation techniques to eliminate unwanted machine noise. However, the emission count rate during AEM has been found to be a good indicator of the damage growth rate in specimens. Furthermore, massive delaminations can be identified with extraordinarily large amplitudes.

Surface temperature monitoring has also proved to be an effective means to detect fatigue damages at the early stages, because heat generation is a consequence of fatigue damage such as delamination and cracking.

Holography is effective as far as detecting delaminations and cracks in the surface plies, both matrix cracks and fiber breaks, and flaws near fine edges. It does not, however, detect subsurface matrix cracking.

Above all, video thermography relates the thermal patterns more directly to the stress field in the material and, hence, is a more appropriate model for studying the mechanical behavior of composite materials.

In addition to currently used techniques and refinements to them, the small angle neutron scattering method (SANS) seems to be another promising technique. The high penetration and selection scattering properties of neutrons provide a powerful capability to study, for example, the changes in the microstructure of bulk specimens. The principal drawback is that the required fluxes are, at present, available only from research reactors. Thus, the method is limited to the study of prototypes rather than in-field examinations.

POTENTIAL OF COMPOSITE MATERIALS

In aircraft structures there has been a remarkable increase in the use of composite materials. These materials offer a considerable weight reduction when compared to conventional metals.

The adaption of such materials to aircraft structures has been limited to secondary structures, such as control surfaces and fairings. The application to primary structures, such as the vertical and horizontal stabilizers, including the wing, remains one of the important research projects to be validated.

The selection, ultimate fabrication and non-destructive testing techniques of these secondary structures led to the confidence to select and build several primary structures. The final stage is, of course, the all-composite aircraft.

In the materials area, emphasis will be on developing and characterizing lower cost material systems, for example, improved epoxy-resin systems with reduced sensitivity to environmental factors, and cost-effective and reliable high temperature resin systems.

Many composite materials today do not fit the production process required to produce an economical and competitive product. In the case of aerospace applications, the most distinguishing characteristics of an advanced composite structure are rigidity, load-bearing capability, and capability to withstand high temperatures. Graphite fibers may very well remain high on the fiber reinforcement list of the future. The polyacrylonitrile-based (PAN) precursor graphite fibers have been the standard for many aircraft applications for years, but the pitch-based fibers offer better processability and a potential cost-break as well. Graphite fibers can be made stronger, also. By alloying graphite fibers with boron, both strength and modulus can be significantly increased.

The demand for advancements in composite materials and economical processes are presented by Goldsworthy (reference 96). He examined the newest composite manufacturing technologies in pultrution and filament winding. From the designer/builder's perspective, Goldsworthy predicted the near future in these manufacturing technologies. In another paper, Kershaw (reference 97) studied two new series of epoxy/resin systems - EPON Resin 9302 and EPON Resin 9310.

As high temperature applications for composites increase, the use of graphite/polyimide will also grow, especially in engine technology. This will be the time to exploit the advantages of applying reinforced composites to the maximum possible extent in a turbine engine.

Composites lack the ductility of metal. Schwartz (reference 84) predicted the potential emergence of fiber-reinforced advance titanium (FRAT), which is a mixture of composite and metallic technologies, to combine the low-cost superplastic-forming diffusion-bonding (SPF/DB) fabrication techniques for titanium with the high strength and stiffnesses of advanced composites.

As manufacturing technologies advance, the demand for non-destructive inspection and examination will increase. The effect of a resin-poor defect is potentially different from that of an impact-damage defect. Therefore, it is essential that non-destructive methods be able to differentiate between these and other types of defects. There are indications, as reported by Schwartz (reference 84), of advances in the state-of-the-art of this technique of non-destructive testing of resin-matrix composites.

Advanced composites still represent a strange new technology to the rest of the American industry. The major driving force here is, of course, lightweight, and the main negative factor is cost. Nonetheless, extensive applications of advanced composites are planned for components on existing airplanes and new airplanes which require structural redesign (references 85, 98).

RESULTS

1. Fundamental work in the area of Composite Material Fatigue/Damage Tolerance continues to provide significant insight into the basic macromechanical behavior of fiber-matrix composites.

2. Presently, there is no precise definition regarding what constitutes fatigue damage of the overall laminate. This is due to the fact there are no established fatigue failure criteria for combined cyclic stresses and the inherent difficulty in predicting lifetime under variable amplitude cycling.

3. The stress-based methodologies and strength degradation models continue to provide insight into the macromechanical behavior of fiber-matrix composites, but more work is needed in the area of damage growth models if this procedure is to become a reliable one for evaluating these materials.

4. It is necessary to include probability theories as well as macromechanics and micromechanics theories when investigating the mechanical behavior of fibrous composite materials.

5. Loss of strength associated with compression stress-oriented impact damage still remains as a major problem regarding fatigue/damage tolerance.

6. Many manufacturers are directing their attention toward developing materials with tougher resin matrices, since delamination is particularly critical when the component is subjected to unanticipated out-of-plane loads.

7. Aircraft manufacturers rely heavily on results found from specimen-component correlations regarding fatigue/damage tolerance simulation.

8. Presently, there are six recognized non-destructive testing methods for testing fibrous composite materials. However, each test method must be selected and tailored to each item, and the geometry of the parts must also be taken into account. For inspection of damage after dynamic fatigue loading, ultrasonic techniques are quite useful. Holography is effective in detecting delaminations and cracks in the surface plies.

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